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Mission-Related Design Requirements for the LEM Propulsion Subsystem

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Apollo Mission Planning Task Force

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Mission-Related Design Requirements for the LEM Propulsion Subsystem [U]

Apollo Mission Planning Task Force [U]

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Approved by
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Report No. LED-540-20

26 Oct. 1964

Approved by A. B. Whitaker,
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1. SUMMARY

The purpose of this report is to define the mission-related critical design requirements for the LEM Propulsion subsystem, and to examine the present subsystem capabilities relative to these requirements for both nominal and contingency situations.

The functional requirements for this subsystem have been listed, together with a list of mission-related parameters which describe these functional requirements. Three of these parameters, thrust magnitude, specific impulse, and ΔV were considered beyond the scope of this study and are accepted. However, the observation is made that it would be desirable to have the LEM propellant tanks sized larger than the loading required for the control weights. This would enable the LEM to operate at greater than its control weight on missions when weight can be transferred from the CSM to the LEM.

The mission-related parameters which have been examined in this report are propellant quantity, helium quantity and flow rate, thrust vector angular range, number of system starts, maximum and minimum times the systems are on and off, and thrust response. In general, the current design values for these parameters are compatible with mission requirements, with the following qualifications:

- Engine burn times can exceed the typical values in the engine specifications. However, the engine capabilities are believed to be within the mission requirements.
- An abort in the powered descent, due to the rupture and loss of fluid from an ascent propellant tank, must be performed at reduced thrust to keep from overpowering the RCS. Rescue by the CSM probably will be required for this contingency.
- An RCS manifold failure after touchdown probably requires rescue by the CSM, due to loss of the ΔV contribution from the RCS to the ascent budget.
- Ascent from the lunar surface after loss of one ascent helium tank is feasible with help from the RCS.
- The three contingencies mentioned above place a requirement on the G&C to complete an ascent, at least to a safe orbit, at reduced thrust. This G&C capability should be confirmed.

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2. LEM PROPULSION SUBSYSTEM DESCRIPTION

2.1 GENERAL

The succeeding paragraphs in conjunction with the Level 2 functional block diagram (see Fig. 1) provide an operational and schematic description of the LEM Propulsion Subsystem.

The LEM has two independent propulsion subsystems - a subsystem for the descent stage which provides variable thrust magnitude and direction, and a subsystem for the ascent stage which provides constant thrust in a fixed direction. Both propulsion systems basically consist of three integrated sections. The first of these is the propellant tank pressurization section which stores and regulates the flow of the high pressure gas that is required for pressurization of the propellant tanks. The second section is the propellant storage and supply section. The rocket engine section then utilizes the propellant flow to provide the desired thrust magnitude. The constituents of each of these sections and a brief operational description of each is presented in the following paragraphs for both the descent and ascent stages.

2.2 DESCENT PROPULSION SUBSYSTEM

2.2.1 Propellant Tank Pressurization Section

As shown in Fig. 1, the propellant tank pressurization section is comprised of high-pressure helium storage tanks, valves and regulators for the control of the flow of helium, and the associated piping.

The gaseous helium, which is stored at 3, 500 psia pressure in each of the two manifolded storage tanks, is isolated from the other subsystem sections prior to the first engine ignition by an explosive squib valve at the manifold outlet. Subsequent to the firing of the squib valve the helium flows downstream through either one of two parallel paths. Each path contains a normally-open latching solenoid shut-off valve followed by a pair of series mounted pressure regulators. In the event of failures, either path may be closed off during engine operation by energizing the solenoid shut-off valve. The pressure settings of the regulators are staggered so that, if the controlling regulator fails open or closed, the downstream regulator or the parallel leg will take over. However, because of the difference in the pressure control settings among the four pressure regulators, one path is always closed off during engine operation. This redundancy provides for normal operation in the event of any regulator failure. After flowing through one of the two parallel flow paths, the pressure regulated helium flow branches out into two flow paths - one that leads to the fuel tanks and the other that leads to the oxidizer tanks. Each of these two paths contains a quad check valve to prevent backflow and mixing of propellant vapors, a burst disc, and a relief valve to prevent over-pressurization of the propellant tanks.

2.2.2 Propellant Storage and Supply Section

The descent stage propellant storage and supply section (see Fig. 1) consists of two fuel tanks, two oxidizer tanks, two filters, and piping. Each set of like tanks contains four interconnecting lines - one each for gas and liquid transfer and one each for gas supply and liquid discharge. The two gas lines enable the tanks to be maintained at equal pressures while the two liquid lines enable both like tanks to empty simultaneously, thus minimizing c.g. movement. The propellant from like tanks which is under the action of helium pressure is forced through the manifolded liquid transfer line through a filter and into the rocket engine section.

2.2.3 Rocket Engine Section

Currently, two rocket engine designs are being developed for the variable thrust descent engine - one by Rocketdyne and the other by STL. Both engines have the capability of varying the thrust magnitude and direction. The main difference between the two is the technique and the hardware with which thrust magnitude is controlled. Both engines are gimballed to provide thrust direction variation. Below is a brief description of each design.

2.2.3.1 Rocketdyne Engine

The Rocketdyne Engine consists of a thrust chamber assembly, a primary propellant valve assembly, a redundant alternate valve assembly, helium control valves, and associated propellant and helium lines. The primary valve assembly regulates propellant flow to the thrust chamber assembly where the propellants react. The redundant alternate valve assembly is used only if the primary set malfunctions.

Each propellant valve assembly includes: a fuel throttle valve, a fuel shutoff valve, an oxidizer throttle valve, an oxidizer shutoff valve, and a servo control valve. In a normal start the throttle and shutoff valves open fully. To throttle, the throttle valves close partly and the shutoff valves remain wide open. The fuel and oxidizer throttle valves are mechanically linked and only the fuel side is driven by an external actuator. The actuator is controlled by the servo control valve which responds to commands from the Stabilization and Control section. At cutoff the throttle and the shutoff valves close completely.

The thrust chamber assembly is composed of propellant manifolds and injector, a combustion chamber, a nozzle and nozzle extension. Fuel and oxidizer coming from the primary valve assembly are distributed across the rear face of the injector, in the injector fuel and oxidizer manifolds. Orifices in the injector determine the direction and velocity of propellant streams entering the combustion chamber. The gases produced by the propellant reaction then flow out of the combustion chamber through an ablatively cooled nozzle which is terminated at an area ratio of 6 to 1 and continue through a radiation cooled nozzle extension to an area ratio of 53 to 1. At thrust levels below 50% maximum thrust, helium is mixed with the propellants in the injector manifolds in order to maintain injector pressure drops commensurate with combustion stability. Fuel and oxidizer helium flows are initiated simultaneously.

2.2.3.2 STL Engine

The STL engine is similar to the Rocketdyne engine except for the method of controlling propellant flow. The propellant flow control valves in this engine consist of a pair of mechanically-linked, variable area, cavitating venturi valves which in turn are mechanically linked to an element in the variable area injector. Thus, the thrust commands that position the propellant flow control valve elements simultaneously position the moving element in the variable area injector.

The thrust chamber assembly is ablatively cooled except for a radiation cooled nozzle extension which begins at an area ratio of 16 to 1 and is terminated at an area ratio of 49 to 1.

2.3 ASCENT PROPULSION SYSTEM

2.3.1 <u>Propellant Tank Pressurization Section</u>

This section consists of two high pressure helium storage tanks that contain about 13.5 lbs. of a helium, explosive squib valves, normally open latching solenoid valves, pressure regulators, quad check valves and relief valves. The helium is stored in two tanks. The tanks are independent. Each has an explosive squib valve. After being filtered and regulated to the proper pressure the helium flows into the Propellant Tank Storage and Supply Section. A relief valve assembly is attached to each tank's helium supply line.

2.3.2 Propellant Tank Storage and Supply Section

The propellant tank storage and supply section for the ascent stage includes one fuel tank, one oxidizer tank and piping. The tanks which are loaded in the oxidizer-to-fuel weight ratio of 1.6 to 1.0, are located at unequal distances from the X axis. This arrangement makes their moment contributions equal so that c.g. offset is minimized. The propellant which is under pressure from the helium within the tanks is available to both the Rocket Engine Section and the Reaction Control System.

2.3.3 Rocket Engine Section

This section basically consists of filters, propellant shut-off valves, valve controls and a thrust chamber assembly.

For redundancy the individual fuel and oxidizer lines that enter the Rocket Engine Section are split into two fuel and two oxidizer lines, each contains a filter, and two shut-off valves in series. Each oxidizer valve is paired with a fuel valve. During start-up, the engine "on" command is transferred to the shut-off valve controls which open all four pairs of shut-off valves simultaneously.

The thrust chamber assembly includes integral propellant manifolds and injector, a combustion chamber, and a nozzle. The manifolds distribute propellant across the rear of the injector. Orifices in the injector direct the propellant into the combustion chamber where the propellants react. The combustion products expand through the nozzle to an area ratio of 43 to 1. The nozzle and chamber are ablatively cooled.

3. MISSION-RELATED DESIGN CRITERIA

3.1 APPLICABLE SPACECRAFT DESIGN GROUND RULES

- The LEM systems shall be capable of meeting their nominal design performance level for a 48 hour mission following separation in lunar orbit. In addition, certain functions will be required during the period from prelaunch to separation.
- The LEM systems shall be designed to accommodate lunar surface day or night environmental extremes.
- No attitude constraints shall be imposed on the LEM due to thermal considerations.
- Wherever possible, a system shall be designed so that the failure of any single element will not cause the loss of a crew member.

3.2 INTERFACE CRITERIA

- The CSM is capable of rescuing LEM from a 50,000 feet clear pericynthion orbit.

4. LEM PROPULSION SUBSYSTEM CRITICAL DESIGN REQUIREMENTS

4.1 FUNCTIONAL REQUIREMENTS

- Two completely independent systems shall be provided: a descent stage subsystem and an ascent stage subsystem.
- The descent stage subsystem shall provide the total impulse required to descend from lunar orbit and hover over the lunar surface. Descent engine thrust shall be variable to provide an efficient descent trajectory and to provide velocity control during the approach to the landing area, hover, and touchdown.
- The descent stage propulsion subsystem shall provide thrust direction control to eliminate thrust moments about the vehicle center of gravity for the normal c. g. range.
- The ascent propulsion subsystem (together with the RCS) shall provide the total impulse required to ascend from the lunar surface to lunar orbit. The ascent subsystem shall be capable of powering an abort to lunar orbit at any time after the initiation of powered descent. Variable thrust magnitude or direction is not required.
- The LEM descent propulsion subsystem shall be capable of providing back-up propulsion for the Service Module Propulsion System (see Reference 1).

4.2 DISCUSSION

4.2.1 Parameter Selection

In an effort to evaluate the functional requirements listed in Section 4.1, the following listed mission-related design parameters were considered:

1. Thrust magnitude
2. Specific impulse
3. Velocity change, ΔV
4. Propellant quantity
5. Thrust level-burn time profiles
6. Helium quantity

7. Helium flow rate
8. Thrust vector angular range
9. Inoperative time
10. Number of system starts
11. Minimum time on
12. Thrust response

The first three parameters; thrust, specific impulse, and velocity change are basic in the design of the propulsion subsystem. All three were established after extensive study by NASA and Grumman. Consequently, it was decided to accept the current values without further study.

Presented in the succeeding paragraphs is a discussion of the mission events and spacecraft considerations that were studied and from which the design requirements are generated. These requirements were obtained by considering the demands upon the subsystem for both nominal and off-nominal missions. The study was carried out within the framework of the previously cited ground rules and functional requirements. Comparison of these critical design requirements with the current LEM design value are made and suggestions for possible improvements indicated. Table I presents in summary form the functional requirements, the associated mission related design parameters, the mission event or profile sizing parameter, the parameter value and the current LEM design value.

4.2.1 Thrust Magnitude

Descent engine thrust must be variable to satisfy the functional requirements. The spacecraft maximum thrust should be determined by the optimum thrust/weight ratio (minimum ΔV) for a suitable descent trajectory. The minimum thrust is determined by the spacecraft weight during hover and touchdown. The established values are accepted as 10,500 lbs. maximum, 1,050 lbs. minimum, although the maximum is no longer optimum.

Ascent engine thrust is determined by weight and trajectory factors as in the case of the descent engine. The established level is accepted as 3,500 lbs.

4.2.3 Specific Impulse

Specific impulse depends on the propellant selection and combustion parameters. The selection of thrust and weight limits indicate the specific impulse range needed.

The descent engine values are:

- at 10,500 lbs thrust, I_{sp} = 305 sec.
- at 1,050 lbs thrust, I_{sp} = 285 sec.
- I_{sp} varies almost linearly with thrust over the given range.

The ascent engine value is 306 sec. All values quoted are the end-of-run 3 sigma minimums.

4.2.4 <u>Velocity Change (ΔV)</u>

The ΔV budget is determined by the selected trajectories, flexibilities, guidance allowances, evaluation uncertainties, etc. The budgets used for this report are described in References 2 and 3 and are summarized in Table I. These budgets have been derived by NASA and GAEC and another detailed study was considered unnecessary. However, this study has resulted in several general conclusions as follows:

- The performance flexibilities provided by the LEM ΔV budget are not large. The flexibilities amount to approximately 4% of the total ΔV budget, and include hover time, provision for line-of-sight to the landing site during powered descent, a brief vertical rise at the beginning of ascent, a $1/2^{o}$ ascent plane change, and docking allowance. Of these, it appears that Apollo performance would benefit the most from an increase in ascent plane change capability. Reference 4 indicates an increase in availability of lunar landing sites if this plane change capability is increased from $1/2^{o}$ to 2^{o}.

- The LEM cannot fly at weights heavier than its control weight, because of propellant tank limitations. This appears to be a disadvantage, in that it is therefore not possible to transfer performance flexibility from the CSM to the LEM when the mission will permit. The point is that all of the CSM performance flexibility, which provides the capability to launch any day of the month to a variety of landing sites, will not be required for every mission. If the choice of lunar landing site and launch date happens to favor the Service Module Performance, a 2-3000 lb increase in LEM weight over the current LEM control weight could be flown within the same total spacecraft weight. However, this performance cannot now be traded from the CSM to the LEM because of the LEM's tankage limitations.

- Therefore, if an increase in the LEM control weight is to be made in the future, consideration should be given to increasing the ΔV available for performance flexibility. Also, consideration should be given to sizing propellant tanks for weights greater than the LEM control weight, so that allowable weight or performance can be traded from the CSM to the LEM when available.

4.2.5 Propellant Quantity, Descent Stage

The required useable propellant quantity for the descent engine is derived from the negotiated ΔV budget shown in Table II. The descent engine ΔV is equal to the total stage ΔV (7385 fps) less a Reaction Control Subsystem (RCS) ΔV allocation of 5 fps for LEM/CSM separation and 4 fps for ullage firing. The net descent engine ΔV is 7376 fps. For a LEM separation weight of 29,870 lbs, the required useable stored propellant requirement is 15,880 lbs. This value is based on an I_{sp} of 301 secs. which is the average I_{sp} during a nominal powered descent trajectory (see Reference 5) and a nominal hover period.

Additional propellant storage for contingencies is not a necessity since the ascent propulsion system is available for aborts if necessary.

The present useable propellant storage capability is 15,920 lbs. which is slightly in excess of that required. Total stored propellant weight must be even greater to provide for possible loading error, off mixture ratio burning, and propellant trapped in the subsystem at burnout. Total LEM propellant storage capability is 16,265 lbs. (Reference 6).

4.2.6 Propellant Quantity, Ascent Engine

The useable propellant is derived from the ascent stage ΔV budget of Reference 3 shown in Table II. The distribution of ascent stage ΔV to the ascent engine and to the RCS is not obvious from Table II. This is because the RCS can make a positive X axis ΔV contribution during powered ascent by accomplishing c.g. moment control with unbalanced couples. The RCS propellant allocation in Reference 3 includes 213 fps for moment control and 17 fps for an ullage firing (provides for a possible two-burn ascent). It can be seen from Table II that the ΔV needed for ascent to 50,000 ft. and insertion into the transfer orbit adds up to 6373 fps. With a 230 fps RCS contribution, the ascent engine must provide 6143 fps.

It should be remembered that if c.g. offset is negligible during ascent, some Guidance and Control (G&C) logic is required to provide an RCS burn either during or at the termination of the ascent engine firing.

The LEM ascent propellant loading is 4,856 lbs. useable at the control weight of 10500 lbs. although the tanks will hold 4,922 lbs. useable. The ΔV available is compared to the requirement below. An I_{sp} of 306 sec. is used.

Lift Off Weight	Useable Propellant	Available ΔV	Req'd ΔV from ascent engine tanks (contingency included)
10,500 lb.	4,856 lb.	6,143 fps	6143 fps
10,566 lb.	4,922 lb.	6,204 fps	

Total propellant storage capability is 5028 lb. with allowances for tolerances.

The ascent tankage is therefore adequate for the ΔV budget when 230 fps is contributed by the RCS. Contingencies involving ascent quantity failures are not considered because of the obvious impracticality of providing additional propellant.

RCS contingencies, however, must be considered in the event that the RCS ΔV becomes unavailable. The critical case is the RCS manifold failure discussed in Reference 7. This failure affects the ascent propulsion in the following manner:

(a) The RCS can no longer provide unbalanced couples because one set of thrusters cannot be used. An RCS ΔV contribution of 213 fps is lost.

(b) In order to keep the c.g. shift from increasing, the ascent main tank propellants are used to feed the RCS. The c.g. shift would come about because only one RCS tank can be emptied and the other remains full. A total of 213 fps is expended from main tanks to provide moment control with balanced couples.

(c) The remaining RCS tank is used to complete ascent by firing in translation mode.

(d) CSM rescue is required if the full ΔV budget allowances for contingency and uncertainties are required. If these prove to be conservative, the LEM may be able to complete rendezvous with this contingency.

The normal and the contingency ΔV allocations are compared in the table below:

	NORMAL	MANIFOLD FAILS
ΔV REQUIRED:		
• To reach 50,000 ft.	6160	6160
• To inject in transfer orbit	113	0
• Contingency	100	0
TOTAL	6373	6160
ΔV AVAILABLE:		
• Main engine tanks	6143	6143
• Unbalanced Couple Moment Control	213	0
• Balanced Couple Moment Control	0	- 213
• RCS Ullage Firing	17	- -
• Available in one remaining RCS tank	- -	251
TOTAL	6373	6181

The ascent propellant quantity is therefore adequate for an RCS failure if LEM is rescued by the CSM, if crossfeed to RCS tankage is used, and if the G&N subsystem is programmed for this thrusting sequence.

4.2.7 Thrust Level - Burn Time Profiles, Descent Stage

Of the thrust level burn time combinations which are possible with the variable thrust descent engine, only those that are critical for rocket design will be discussed. Thus, the following combinations were determined by considering that all the stored useable propellant (15,880 lb.) in the descent stage is burned. All calculations are based on LEM separation weight of 29,870 lb.

The maximum thrust time profile was generated by assuming all start-ups at full thrust (no c. g. misalignment) and an abort with the descent engine from the maximum thrust phase of a nominal powered descent trajectory (see Ref. 5). All fuel is consumed during a

[redacted]

9 second transfer orbit insertion and 448 seconds of descent and abort - - a total of 457 seconds at 10,500 lbs. thrust.

The derivation of low thrust level burn time profiles are based upon two extreme visibility phase cases during the powered descent. It should be noted that time at low thrust is not equivalent to time at high thrust. Therefore, times are derived for two thrust ranges, less than 50% and near 100%.

The first case assumes that there is no necessity for a visibility phase. In this case, the LEM goes from the 50,000 ft. pericynthion to the hover point in one burn at high thrust. During this first burn the minimum ΔV expenditure is 5,838 fps at high thrust. The burn time for this phase amounts to 383 seconds excluding the start-up time. Expenditure of the remaining fuel during hover results in an additional 274 seconds of burn time. Thrusting at 10% (1,050 lbs) for 30 seconds occurs at each start up to allow for thrust vector alignment with the c.g. Hohmann transfer orbit insertion requires an additional 6 seconds at full thrust. Total throttled burn time is 334 sec and total full thrust burn time is 389 sec.

The second low thrust level profile is generated by maximizing the time spent in the visibility phase and minimizing the hover time. The use of lower thrust during the first (high thrust) phase of powered descent is not considered because of the penalty associated with low thrust-to-weight ratio and lower I_{sp}. The minimum ΔV necessary during the high thrust phase is 4947 fps and the burn time is 345 seconds excluding the start-up time. The minimum ΔV for hover and touchdown is 336 fps from Reference 8 and the resulting burn time is 60 seconds. Thus, the ΔV available for the visibility phase is 1724 fps allowing for the T/W ΔV penalties and the lower I_{sp}. The burn time during this phase is 183 seconds. Thrusting at 10% for 30 sec. occurs at each start up and full thrust for 6 sec. is required to complete the Hohmann transfer orbit insertion. Total burn times are 303 seconds throttled and 351 seconds at full thrust.

The LEM propulsion design specifications do not establish maximum burn times. A table showing a typical mission is given in the specifications (Reference 9 and 10). The data are given below.

	Thrust Level (lbs)	Burn Time (secs.)
Acceptance Tests	1,050 to 10,500	135
Insertion & Descent	10,500	415
Visibility Phase	5,750	60
Hover to Touchdown	5,750 to 1,050	120

[redacted]

The required burn times for the various critical design missions and those of the Design Reference Mission are summarized for comparison in the table below.

REQUIRED BURN TIMES

	Maximum Thrust Mission (Abort)	Minimum Visibility Descent Mission	Maximum Visibility Descent Mission	Design Reference Mission
High Thrust, Approx. 10,500 lb.	457	389	351	360
Low Thrust, Less than 5,750 lb.	0	334	303	244

In some instances the descent engine times for the critical design mission exceed those presented in the specifications. These deviations however, are not, considered significant and are believed to be within the descent engine capabilities. Accurate burn time capability can only be predicted after the completion of the engine development program. In any event, time can be borrowed from the acceptance test allowance.

4.2.8 Thrust Level - Burn Time Profile, Ascent Engine

The thrust level burn time profile for the ascent engine is derived by consuming all the ascent stage fuel at the design operating thrust level of 3500 lbs. For 4922 lbs. of maximum stored useable propellant the burn time is 430 sec. based on an I_{sp} of 306 seconds.

The typical mission value in the design specification (Reference 11) is 385 seconds of operation at 3500 lbs. thrust. The pre-launch acceptance test time allowed is 60 seconds. No maximum time is established in the specification (Reference 11).

It is believed that engine capability will be sufficient to accomplish the 430 second burn.

4.2.9 Helium Quantity, Descent Engine

The helium quantity required for the descent engine is that needed to expel 15,880 lbs. of useable propellant at constant nominal tank pressure. That amount is 46.2 lbs. It includes approximately 12 lbs. of unuseable helium. Constant ullage pressure is needed for the STL cavitating venturi flow control valves which require constant inlet pressures during throttled operation.

On the other hand, the Rocketdyne engine tank pressures can be allowed to decay during throttled operation. This method of operation is termed the blowdown mode; it takes place at thrust levels below 50%. Helium is needed from the tank supply lines to satisfy the injection requirements of the Rocketdyne engines; therefore helium for tank pressurization is sacrificed. The normal mission which required the greatest amount of helium for the Rocketdyne engine is that with maximum hover time (i.e., no visibility phase). For this mission, the helium requirement is 41.6 lbs. based upon the requirements of 10500 lbs. of thrust for 383 seconds during powered descent.

The contingency which requires the greatest amount of helium for the Rocketdyne engine (43.9 lbs.) consists of an abort utilizing the Abort Guidance Subsystem (AGS) at the end of the maximum visibility phase of a descent trajectory. The thrust level requirement for aborts during this phase was assumed to be 7600 lbs. and was assumed to last for the time it takes to consume all remaining fuel. The actual thrust level requirement will vary from 7600 lbs. to 6600 lbs. for aborts from the visibility phase utilizing the AGS, but the difference in helium quantity is negligible.

Consequently, the requirement for the STL engine to operate at constant ullage pressure while all fuel is expelled, constitutes the sizing criteria for the descent stage helium tanks. This requirement is smaller than the current LEM capacity which is 48.6 lbs. based on 16,256 lbs. of expelled propellant.

Provisions for redundant helium tanks are unwarranted since the ascent propulsion system is available for aborts.

4.2.10 Helium Quantity, Ascent Stage

The useable stored propellant quantity is 4922 lbs. The ascent stage helium quantity required to maintain constant nominal ullage pressure during the consumption of all the propellant is 13.5 lbs. A redundant helium supply is desirable, but complete redundancy is unnecessary. A reasonable single failure criterion is to have enough helium remaining to obtain a safe orbit and await CSM rescue.

If the 13.5 lb. nominal requirement is stored in two independent tanks, one of those tanks could expel the entire propellant supply, if ullage pressure and therefore thrust are allowed to decay (blowdown mode). The ΔV available in this mode from 4856 lbs. of main tank propellant and 168 lbs. of RCS propellant (moment control) is 6097 fps. A ΔV of 6190 fps (6160 from budget plus 30 fps estimated for longer burn) is needed to reach 50,000 ft. circular orbit on primary guidance. The deficiency is only 93 fps which is easily supplied from the 290 fps remaining capability of the RCS.

Current LEM ascent helium useable storage capability is 13.5 lbs. which is adequate for the above criteria.

4.2.11 Helium Flow Rates, Descent Stage

The maximum flow rate is based upon the propellant flow required for 10,500 lbs. of thrust for either the Rocketdyne or STL Engines. This helium flow is 4.50 lbs/min. The required helium flow rate to meet contingencies is the same. Current LEM capability is sufficient and is 22.50 lb/min at start-up decreasing to a maximum of 4.50 lbs/min after nominal propellant tank pressures are achieved.

4.2.12 Helium Flow Rates, Ascent Stage

Ascent Stage maximum nominal and contingency helium flow rate requirements are the same and are based upon the propellant flow required for 3,500 lbs. thrust. The amount is 1.30 lbs/min. Current LEM capability is 6.50 lbs/min at start-up with the flow decreasing to 1.30 lbs/min after nominal propellant tank pressures are achieved.

4.2.13 Descent Engine Thrust Vector Angular Range

The normal angular range of thrust vector is determined by the most offset c.g. location for a normal mission. This could occur prior to injection of the LEM into the coasting descent trajectory. It is assumed in this case that the propellants transfer so that the ullage of one oxidizer or fuel tank has been completely filled with oxidizer or fuel from the corresponding second tank. The possible oxidizer transfer is approximately 140 lbs. at control weight. Similarly, about 85 lbs. of fuel is assumed to have shifted. These transfers result in a c.g. approximately 1.0° from the nominal thrust vector.

The contingency which causes maximum c.g. shift is an ascent stage oxidizer tank rupture (an instantaneous loss of approximately 3,000 lbs. of oxidizer) during the high thrust phase of the powered descent. This results in a c.g. which is at least 10° out of line with the thrust vector. The RCS cannot handle the moment unbalance (approximately 4,500 ft-lb) caused by this failure unless a reduction in thrust reduces the moment unbalance to RCS capability (i.e., 1100 ft-lbs for 2 jet couple).

Figure 2 shows the maximum allowable thrust levels (within RCS capability) for aborts from powered descent that are caused by the aforementioned failure. It is possible to abort as late as approximately 300 seconds after the initiation of powered descent and yet accomplish a successful abort by burning all the remaining descent stage propellant. Aborts later than approximately 300 seconds are not possible because of the lack of descent stage propellant.

Current LEM thrust vector angular range is $\pm 6^{\circ}$ which provides the minimum required 1.0° and also allows for possible propellant off loading and uncertainties in engine mounting, and variations in the gas flow within the thrust chamber. It can be seen, however, that this capability is insufficient to satisfy the $\pm 10^{\circ}$ contingency requirement. To satisfy this requirement one of two things must be done.

1. The thrust vector angular range must be increased to $\pm 10^{\circ}$.
2. The descent engine must be operated at reduced thrust during aborts from powered descent.

At present the first choice appears to require both rocket engine and LEM structural re-design. The second choice, appears more feasible since it is within present structural and propulsion system capability, although additional requirements are imposed on the G&C Subsystem. Further evaluation of the effects would have to be made before a final selection is made.

4.2.14 Maximum & Minimum Time Off, Descent Stage

The maximum time that the descent stage is shut down between firings is about three hours. This occurs in the case of inserting into the coasting descent trajectory, continuing around the moon (in a reconnaisance orbit), and firing the descent engine again for descent at the pericynthion.

Minimum time off is almost negligible for the case of an immediate abort following coasting descent trajectory insertion. Times are not mentioned in the design specifications, but there is no apparent limitation on minimum time off.

4.2.15 Maximum & Minimum Time Off, Ascent Stage

Maximum time off between firings for the ascent engine is about nine hours. This is obtained after a 9 hour, 50,000 ft. orbital contingency period. Minimum time off may be negligible (back-to-back firing) as in the case of late launch at the limit of the launch window. Theoretically, in this case, some small parking time is required in a 50,000 ft. orbit. Again, times are not mentioned in the design specifications, but there is no apparent limitation on minimum time off.

4.2.16 <u>Number of Starts, Descent Engine</u>

Normally, there are two engine starts during the mission. The first firing is for the injection into a Hohmann type coasting descent trajectory. This firing is followed by another approximately one hour later. This last firing completes powered descent.

In a contingency three more starts may be required. An abort early in powered descent on the AGS may require that the descent engine perform one midcourse correction and one rendezvous burn (because of the larger ΔV requirements when AGS is used). If such is the case, the descent engine will be put through a total of four firing cycles.

The engine specifications require capability for 20 starts (References 9, 10).

4.2.17 <u>Number of Starts, Ascent Stage</u>

Normally, there is only one ascent engine start. The firing begins on the lunar surface and continues until the LEM is placed in the nominal coasting ascent trajectory with proper burnout conditions. The contingency of a lunar launch using the AGS may require four firing cycles as follows: powered ascent, one plane change, one midcourse correction and one rendezvous firing.

The specification calls for capability of 35 starts (Reference 11).

4.2.18 <u>Minimum Time On</u>

The minimum normal time on for the descent engine is about 8 seconds and occurs during full thrust insertion into the coasting descent trajectory. There is also the possibility of using the descent engine for midcourse or rendezvous firings during aborts with the descent engine in which case burn times would be of 2 - 4 second duration.

For late launch, firing time for the ascent engine is 5 seconds during insertion into coasting ascent after a 50,000 ft. parking orbit. During ascent, the ascent engine could also be used for midcourse or rendezvous burns as short as 2 - 4 seconds in the event of loss of an RCS thruster manifold. Minimum on times are not mentioned in the design specifications, but there are no known engine limitations in this respect.

4.2.19 <u>Thrust Response, Descent Engine</u>

The guidance system can sense and compensate for the impulse accumulated during the thrust rise, but the impulse at shutdown ("tail-off") should be predictable and repeatable.

The minimum thrust rise time should allow the engine to reach a stable thrust level, from which the tail-off impulse can be predicted. The minimum descent engine impulse requirement has been estimated to be equivalent to about 2 seconds of full thrust burning. Also, the tail-off impulse is estimated to be equivalent to much less than 1 second of full thrust. Therefore, it appears that a rise time of less than 1 second will meet the minimum impulse requirements.

A minimum shutdown response time is not critical for a normal mission. The shutdown response requirement is generated by the time required to stage and abort near the lunar surface. A shutdown time of 0.5 seconds or less is compatible with the vehicle abort staging response time.

Thrust response to throttle command must be commensurate with the G and C Subsystem during powered descent and both automatic and manual modes of hover-to-touchdown. In addition, a rapid thrust reduction is necessary for the contingency discussed under Thrust Vector Angular Range where ascent propellant is suddenly lost during powered descent.

The determination of the required impulse repeatability and thrust response requires comprehensive dynamic analysis. The necessary studies have not been completed. It is possible, however, to estimate a possible tolerance for shutdown impulse which is probably most critical after a Hohmann transfer orbit insertion. If the allowable error in pericynthion altitude is 5000 ft., allowable ΔV error is 2.3 fps which is equivalent to .20 second burn time at 10,500 lb. thrust.

The descent engine specification lists the following capability which appears sufficient:

a) For start, 0 to 90% thrust within 0.25 sec.

b) For shutdown, 100% to 10% thrust within 0.25 sec.

c) For thrust response, 100% to 10% thrust within 1.0 sec. and 100% to 50% thrust within 0.5 sec.

It is assumed that the abort guidance system will not impose more severe engine thrust response requirements than the primary system.

4.2.20 Thrust Response, Ascent Engine

Start-up and shutdown times are subject to the same considerations discussed under Descent Engine. Additional considerations, however, are abort from hover near the surface and heat from the exhaust gases during the separation from the descent stage.

Abort from hover near the surface is sometimes required if the descent engine fails or is shut down prematurely. If altitude and descent rate are low enough, it is possible to drop in with an impact velocity less than the design limit (10 ft/sec). If drop-in is not permitted, the descent section is staged and the ascent engine is started. The descent rate should be arrested so that the abort is performed without touching the impacted descent stage. The allowable combinations of indicated altitude and descent rate during final descent will depend on the time to complete all sequences, the thrust-weight ratio and the accuracy of measuring altitude and descent rate.

Figure 3 shows the above abort boundaries for the assumptions stated in the figure. The engine start-up time is .25 seconds and is initiated .25 seconds after descent engine shut off. It is seen that no safe approach corridor exists. If instantaneous thrust is assumed, the safe abort boundary is still not displaced enough to allow a safe corridor. Therefore start-up times of less than .25 seconds are not essential.

Other considerations include the temperature limits during staging. The ascent stage is attached to the descent stage when the ascent engine is ignited and the exhaust gases are deflected overboard by a structural shield on the descent stage. Current shield design is for .5 sec. maximum thrust build up time. Extended build up times could also damage the nozzle extension of the engine.

Current LEM capability as delineated in the specification, appears sufficient and is as follows:

a) For start-up, 0 to 90% thrust within 0.20 sec.

b) For shutdown, 100% to 10% thrust within 0.20 sec.

5 CONCLUSIONS AND RECOMMENDATIONS

In general, the present mission - related requirements to which the LEM propulsion system is being designed are adequate to meet all anticipated nominal and contingency situations. Specifically, the following conclusions and recommendations are made:

- The ΔV budget for the LEM contains a minimal allowance for performance flexibilities. Further, current tank sizing limits the ability to transfer performance capability from the CSM to the LEM, when the mission would allow this to be done. If the LEM control weight is to be increased in the future, consideration should be given to increasing the ΔV for flexibilities, and to sizing the tanks for capacities greater than required for the control weight.
- Engine burn times can exceed the typical values in the engine specifications. However, the engine capabilities are considered adequate for the missions defined. This conclusion should substantiated by ground test firings simulating the mission shown.
- An abort in the powered descent, due to the rupture and loss of fluid from an ascent propellant tank, must be performed at reduced thrust to keep from overpowering the RCS. Rescue by the CSM probably will be required for this contingency.
- An RCS manifold failure after touchdown probably requires rescue by the CSM, due to loss of the ΔV contribution from the RCS to the ascent budget.
- Ascent from the lunar surface after loss of one ascent helium tank is feasible with help from the RCS.
- The three contingencies mentioned above place a requirement on the G & C to complete an ascent, at least to a safe orbit, at reduced thrust. This G & C capability should be confirmed.

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10. Berman, Gardner and Dandridge, Engine, Rocket, Liquid, Variable Injector, Descent, Design Control Specification for, GAEC LSP-270-6, 29 June 1963.

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TABLE I PROPULSION SUBSYSTEM
REQUIREMENTS SUMMARY

Subsystem Function	Parameters Which Describe Function	Mission Event or Profile-Sizing Parameter	Current Design Value
To Provide Impulse	Descent Stage propellant quantity	Normal: 15,880 lbs. based on a budgeted design engine ΔV requirement of 7376 fps for LEM separation weight of 29,870# and for a specific impulse value equivalent to the average over the nominal thrust profile (i.e., Isp = 301 seconds) Contingency: None	15,920 lb.
	Ascent Stage propellant quantity	Normal: 4856 lbs. based on an allocated ascent stage main engine ΔV 6143 fps at the control weight of 10,500 lbs. and a specific impulse of 306 sec.	4922 lb. full 4856 lb. allowed for control weight.
Provide Thrust Vector Magnitude	Descent Stage thrust level-burn time profile	Normal: 389 sec. at high thrust and 334 sec. at low thrust during minimum visibility descent Contingency: 457 sec. at high thrust during abort from powered descent.	typical times: 415 sec. at high thrust 180 sec. at low thrust plus 135 sec. for acceptance test.
	Ascent Stage thrust level-burn time profile	Normal: 430 seconds to consume all propellant during ascent from lunar surface. Contingency: None	385 sec. typical time plus 60 sec. for acceptance tests.

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TABLE I PROPULSION SUBSYSTEM REQUIREMENTS SUMMARY (Cont.)

Subsystem Function	Parameters Which Describe Function	Mission Event or Profile-Sizing Parameter	Current Design Value
Provide Thrust Vector Magnitude	Descent Stage helium quantity STL Engine	Normal: 46.2 lbs. of helium based on expelling all useable propellant (i.e., 15,880 lbs) at the nominal ullage pressure. Contingency: None	48.6 lbs.
	Rocketdyne Engine	Normal: 41.6 lbs. of helium based on a maximum hover condition with no visibility phase. Contingency: 43.9 lbs. of helium based on an abort on AGS from the end of the visibility phase with all fuel consumed.	
Provide Thrust Vector Magnitude	Ascent Stage helium quantity	Normal: 13.5 lbs. of helium based on expelling all useable propellant (i.e., 4922 lbs) at the nominal ullage pressure Contingency: 13.5 lb. in two redundant tanks allows ascent after tank failure with thrust decay.	13.5 lbs. in two tanks.
	Descent Stage helium flow rates	Normal: 4.50 lbs/min based on a max. thrust of 10,500 lbs. Contingency: Same	22.50 lbs/min at start-up which decreases to 4.50 lbs/min after stabilization.

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TABLE I PROPULSION SUBSYSTEM REQUIREMENTS SUMMARY (Cont.)

Subsystem Function	Parameters Which Describe Function	Mission Event or Profile-Sizing Parameter	Current Design Value
Provide Thrust Vector Magnitude	Ascent Stage helium flow rates	Normal: 4.50 lbs/min based on a max. thrust of 3,500 lbs. Contingency: Same	6.5 lbs/min at start-up which decreases to 1.30 lbs/min after stabilization.
Position the descent engine thrust vector through the c.g.	Descent engine thrust vector angular range	Normal: $\pm 1^{\circ}$ based on a c.g. shift after separation in lunar orbit due to ullage shift of both fuel and oxidizer. Contingency: At least $\pm 10^{\circ}$ based on a c.g. shift due to loss of one fuel or oxidizer tank in the ascent stage during the high thrust portion of the powered descent.	$\pm 6^{\circ}$
Thrust Control	Descent Stage maximum and minimum time-off	Normal: 3 hours max. and 1 hour min. 1 reconnaisance orbit plus coasting descent to pericynthion descent transfer orbit. Contingency: 2 hours max. and 0 hours min. based on an abort prior to the initiation of powered descent and for an immediate abort following insertion into the coasting descent orbit.	Typical times: 2 hours max. and 1/2 hr. min.

TABLE I PROPULSION SUBSYSTEM

REQUIREMENTS SUMMARY (Cont.)

Subsystem Function	Parameters Which Describe Function	Mission Event or Profile-Sizing Parameter	Current Design Value
Thrust Control	Ascent Stage maximum & minimum time-off	Normal: 0 time for both max. and min. based on a normal ascent from the lunar surface. Contingency: 9 hours max. and 0 hours min. for a 9 hour orbital contingency and for a late launch at the late limit of the launch window.	28 hours typical time off
	Descent Stage number of starts	Normal: 2 req'd for descent orbit insertion and powered descent. Contingency: 4 req'd for descent orbit insertion and abort from powered descent using descent engine for abort, midcourse correction and rendezvous.	15 firing cycles exclusive of an additional allowance of 5 for testing.
	Ascent Stage number of starts	Normal: 1 req'd for lunar launch through transfer orbit injection. Contingency: 4 req'd for ascent from lunar surface on AGS, a plane change, midcourse correction and rendezvous.	30 firing cycles exclusive of an additional allowance of 5 for testing.
	Descent Stage Minimum Time On	Normal: 8 seconds, insertion into coasting descent trajectory. Contingency: 2 seconds, abort trajectory midcourse corrections.	Not specified

TABLE I PROPULSION SUBSYSTEM REQUIREMENTS SUMMARY (Cont.)

Subsystem Function	Parameters Which Describe Function	Mission Event or Profile-Sizing Parameter	Current Design Value
Thrust Control	Ascent Stage Minimum Time On	Normal: 5 seconds, insertion into coasting ascent after a 50,000 ft. parking orbit. Contingency: 2 seconds, for midcourse or rendezvous, in the event of RCS malfunction.	Not specified
	Descent Stage start up & shut-down time.	Normal: None Contingency: 1 sec. start-up from consideration of 2 sec. burn time. 0.5 sec. shutdown for stage and abort near lunar surface	0 to 90% thrust within 0.25 sec. for all start ups; 100% to 10% thrust within .25 sec. for all shutdowns.
	Ascent Stage start up & Shutdown time.	Normal: 0 to 90% within 0.5 sec. for staging on lunar surface. Contingency: 0 to 100% within .25 sec.	0 to 90% within 0.2 sec. 100% to 10% within 0.2 sec.
	Descent Stage thrust response	Normal: See text Contingency: None	100% to 50% within 0.5 sec., 100% to 10% within 1.0 sec.

TABLE II

LEM MINIMUM ΔV BUDGET (REFERENCE 3)

DESCENT STAGE

BUDGET ITEM / MISSION PHASE	OPEN LOOP: ABSOLUTE MINIMUM	OPEN LOOP: FLEX. ALLOWANCE	GUIDANCE	FLIGHT MECHANICS CONTINGENCIES	EVAL. UNCER.	TOTAL
LEM Separation	5					5
Descent Transfer Orbit Insertion	97		2		1	100
Descent to Surface						
. Initial Deboost	5785	165*	120		60	6130
. Landing Approach Translation and Touchdown		900			250	1150
TOTAL ΔV						7385

ASCENT STAGE

Lunar Launch						
. To 50,000 ft.	5880	100**	120	100	60	6260
. Ascent Transfer Orbit Insertion	100	10***	2		1	113
Midcourse Corrections			50			50
Rendezvous	97	89	10		2	198
Docking		25				25
TOTAL ΔV						6646

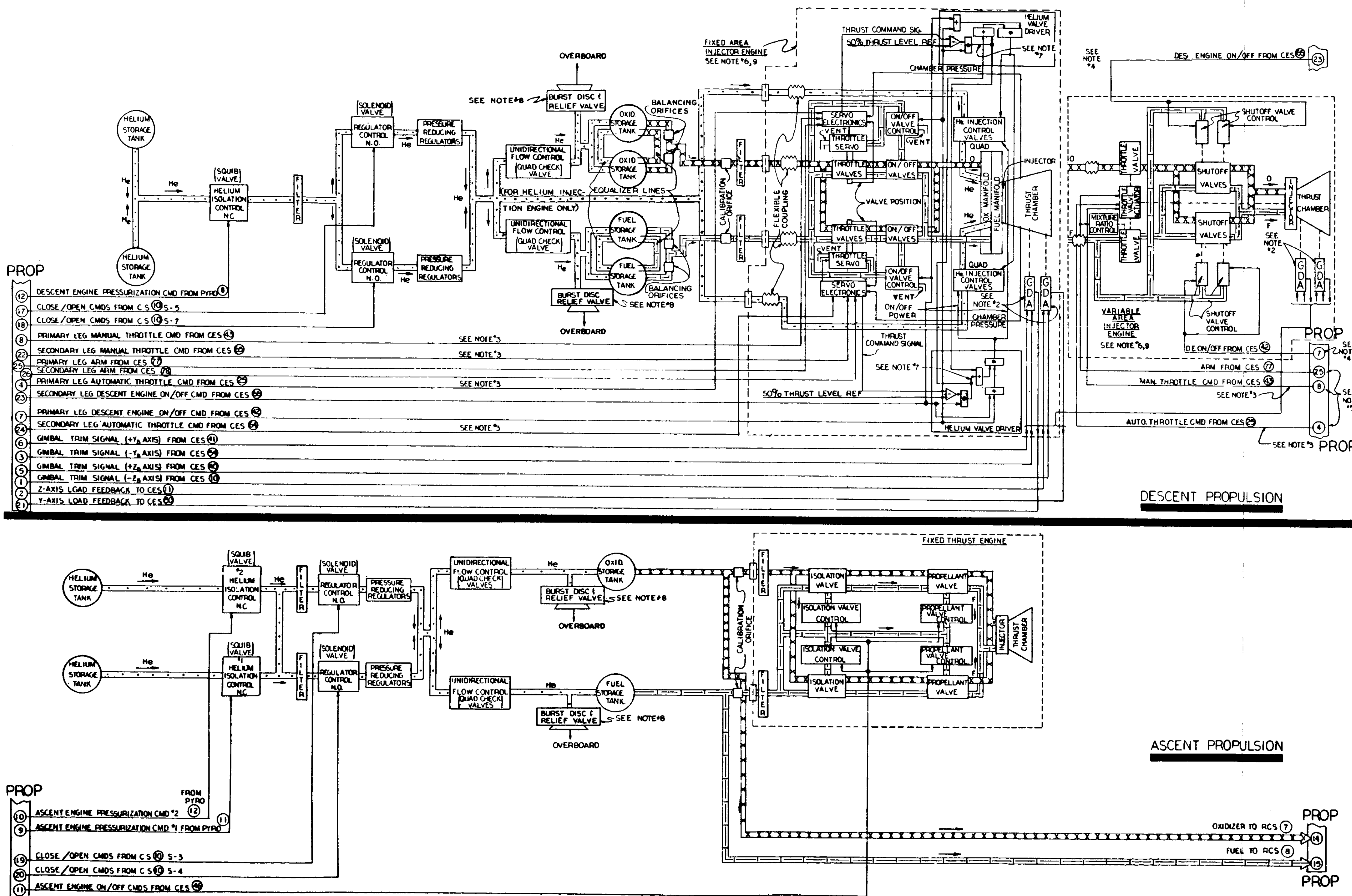
* Shaped for pilot visibility

** 12 second vertical rise and finite pitch rate requirements

*** $1/2^{\circ}$ plane change and non-Hohmann transfers

MISSION PARAMETERS

1. CSM in 80 n. mi. altitude circular parking orbit.
2. LEM powered descent initiated at 50,000 feet.
3. (T/W) at pericynthion = 0.356
4. (T/W) at lunar lift off = 0.333
5. Separation Wt = 29870 lbs., Liftoff Wt = 10,500 lbs.



This diagram is the same as the Level 2, LEM Basic Functional Configuration Propulsion Subsystem, LDW-540-10008, Rev. A

NOTES

1. CONTROL PANEL SUBASSEMBLY DELETED. ALL SIGNALS FROM CONTROL PANEL ARE NOW PART OF FUNCTIONAL INTERFACE TO AND FROM CREW SYSTEMS (CS).
2. GIMBAL DRIVE ACTUATOR PART OF STABILIZATION & CONTROL SUBSYSTEM
3. THE USE OF SEPARATE AUTOMATIC AND MANUAL THROTTLE COMMANDS TO THE DESCENT ENGINE IS CURRENTLY UNDER STUDY BY PROPULSION.
4. THERE IS NO PRIMARY OR SECONDARY SELECTION REQUIREMENT FOR THE VARIABLE AREA INJECTOR ENGINE. THE CES WILL PROVIDE BOTH PRIMARY AND SECONDARY ENGINE ON COMMANDS WHEN USED WITH THIS ENGINE.
5. IN THE CASE OF THE VARIABLE AREA INJECTOR DESCENT ENGINE BOTH ENGINE ARM POWER AND THROTTLE COMMANDS WILL BE PROVIDED TO THE ENGINE INDEPENDENTLY OF ENGINE ON COMMAND
6. THERE ARE TWO DES. ENGINE CONFIGURATIONS UNDER CONSIDERATION FOR USE IN THE LEM. HELIUM INJECTION IS NOT REQUIRED FOR THE VARIABLE AREA INJECTOR ENGINE. THE VARIABLE AREA INJECTOR ENGINE MAY BE INTERFACED TO THE OXIDIZER AND FUEL LINES AT POINTS O AND F.
7. THE COMPARATOR WILL PROVIDE AN OUTPUT WHEN THE THRUST COMMAND SIGNAL REPRESENTS A THRUST LEVEL OF 50% OR LESS. THIS FUNCTION APPLIES TO THE FIXED AREA INJECTOR ENGINE ONLY.
8. BURST DISC AND RELIEF VALVE VENT THE He OVERBOARD TO PROVIDE NO NET THRUSTING EFFECT (NO THRUST COUPLES RESULT).
9. DESCENT ENGINE IS GIMBALLED ABOUT THE Y & Z BODY AXIS.

ABBREVIATIONS

- CES = CONTROL ELECTRONICS SECTION
- CS = CREW SYSTEMS
- RCS = REACTION CONTROL SUBSYSTEM
- PYRO = PYROTECHNICS
- Y_B = BODY Y-AXIS
- Z_B = BODY Z-AXIS
- GDA = GIMBAL DRIVE ACTUATOR
- CMDS = COMMANDS
- DES = DESCENT
- MAN = MANUAL
- AUTO = AUTOMATIC
- O = OXIDIZER
- F = FUEL
- He = HELIUM

SYMBOLS

VALVE POSITION INDICATOR

COMPARATOR (SEE NOTE *7)

A, B, C FUNCTIONAL 'AND' GATE — FUNCTIONAL INPUTS A AND B WILL PROVIDE OUTPUT C. THIS FUNCTIONAL GATE IS NOT DESIGNED FOR '0', '1' LOGIC

A, B, C FUNCTIONAL 'OR' GATE — FUNCTIONAL INPUTS A OR B WILL PROVIDE OUTPUT C. THIS FUNCTIONAL GATE IS NOT DESIGNED FOR '0', '1' LOGIC

OXIDIZER

FUEL

HELIUM

Fig. 1 Propulsion Subsystem Functional Diagram

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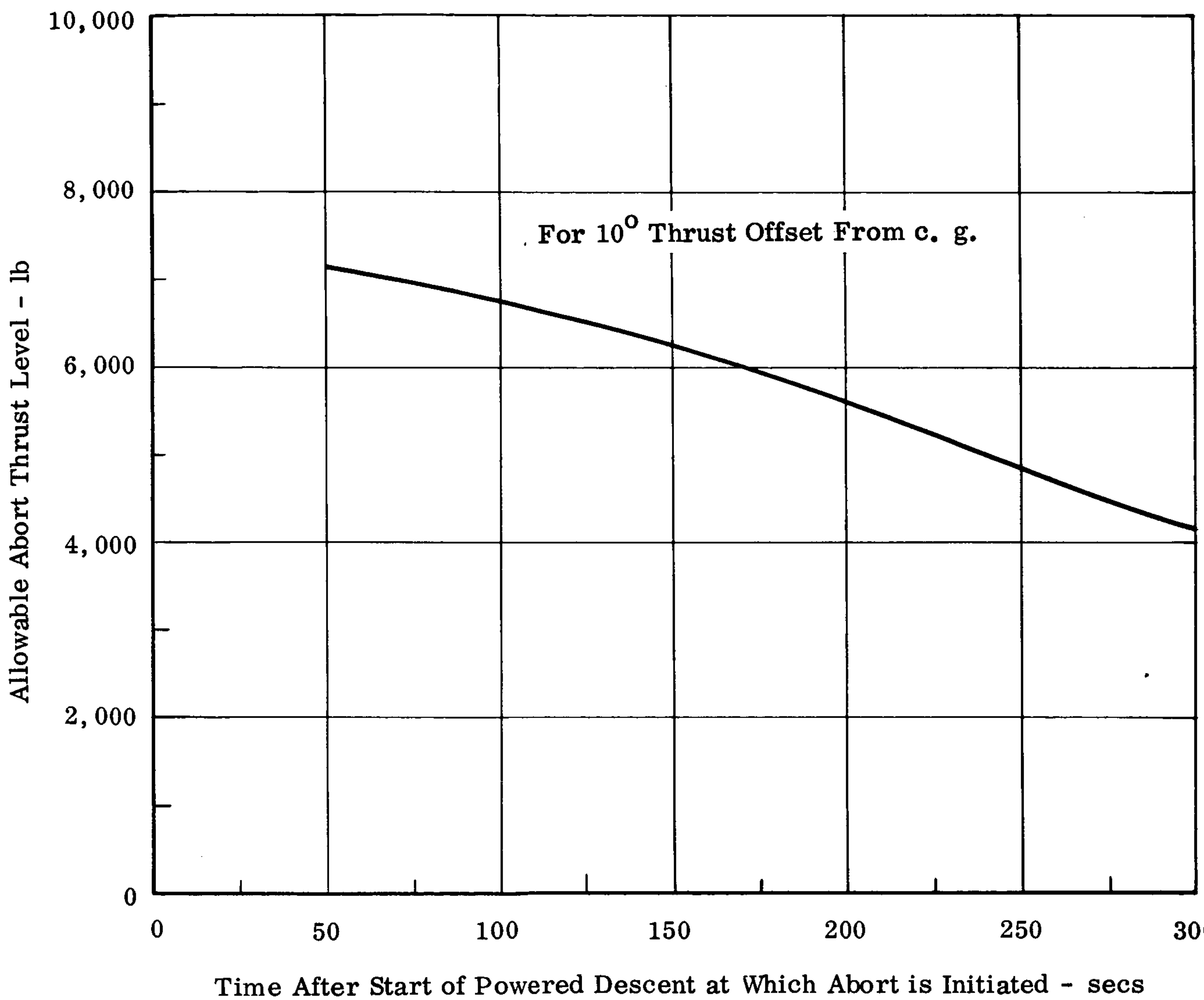

Fig. 2 Maximum Allowable Thrust Levels for Aborts from Powered Descent Due to Ascent Stage Oxidizer Tank Failure

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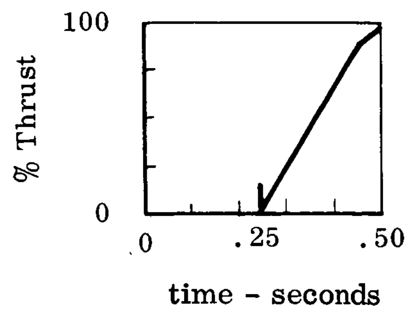

Assumed Thrust Profile

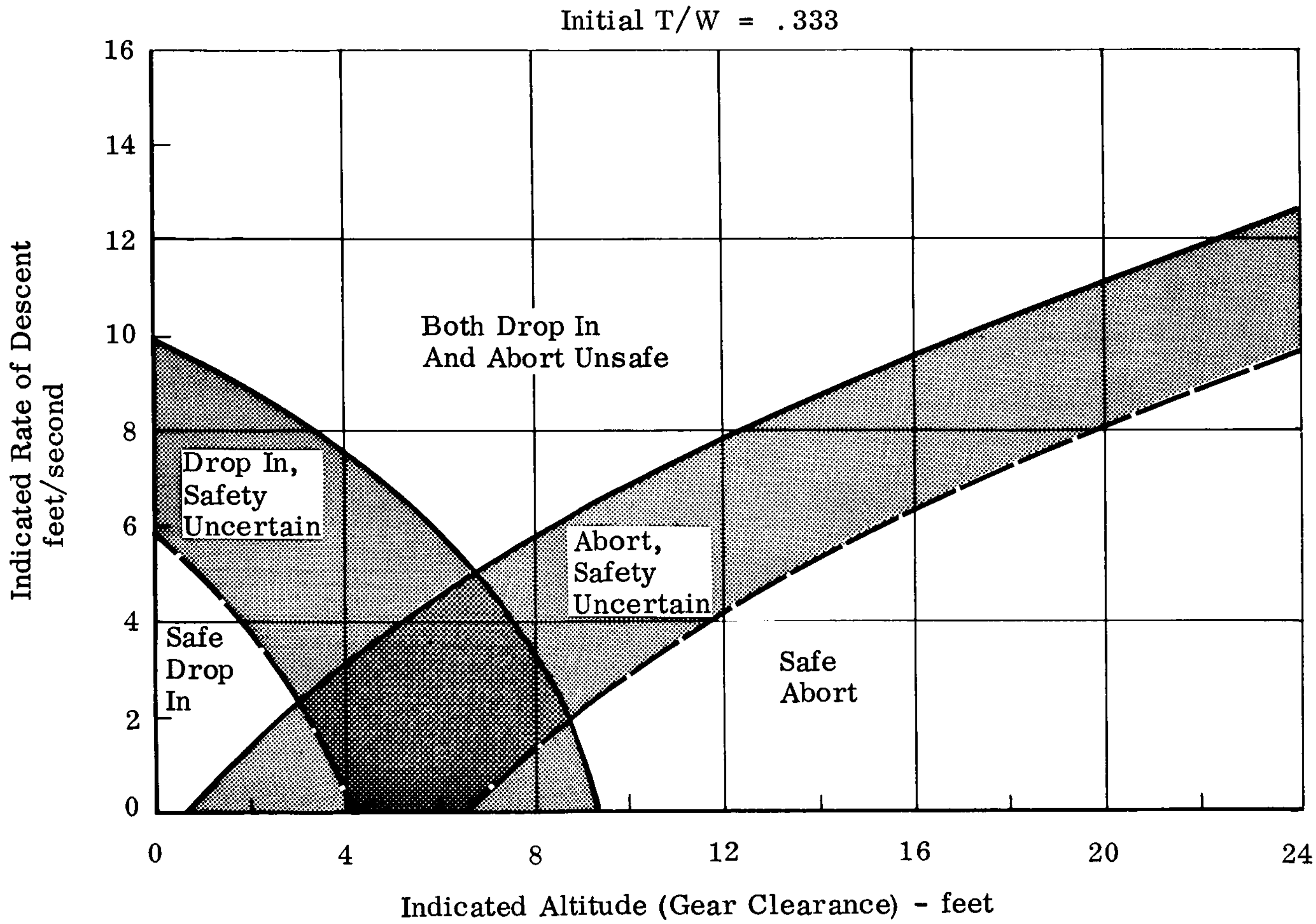

Fig 3 Aborting Near the Lunar Surface Because of Premature Descent Engine Shutdown